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APPLICATIONS OF LOW-POWER NUCLEAR ROCKETS

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INTRODUCTION

The most challenging use of nuclear rockets is for manned interplanetary flight. The high specific impulse of these propulsion systems gives them a large advantage over chemical rockets. With nuclear rockets it is possible to make fast round trips with reasonable weight (refs. 1 to 3). For example, studies indicate that manned chemical rocket vehicles for similar missions would weigh as much as 10 times more than nuclear rockets. With such gains in mind, the United States program of nuclear rocket development is aimed toward manned interplanetary flight.

Prior to, or in addition to, operations of this magnitude there may be other uses of nuclear rockets. Four such uses are as follows:

- (1) As the last stage of manned interplanetary missions
- (2) Pioneering space probes
- (3) Space exploration beyond the capability of manned vehicles
- (4) Perigee propulsion of manned and unmanned vehicles

The nuclear rockets required for these missions require about an order of magnitude less power than manned interplanetary missions. These missions are therefore called low-power nuclear rocket missions (refs. 4 to 7). For example, in the case of the final stages of manned interplanetary vehicles the power levels are about 1000 megawatts or 50,000 pounds of thrust, instead of 10,000 megawatts and 500,000 pounds of thrust, which is required for leaving Earth orbits with manned nuclear vehicles.

The purpose of the pioneering space probes would be to determine the environment man will be exposed to on interplanetary voyages and to gather data useful for the planning of protective measures. Unmanned probes for scientific exploration of space would include such missions as (1) those to the more distant planets and extremities of the solar system, (2) trips close to the sun where it would be impractical to provide protection for men, and (3) any other trips that cannot be accomplished if the requirement of return to Earth is added to the mission.

The fourth use of low-power nuclear rockets would be in connection with the perigee propulsion of manned and unmanned vehicles. Perigee propulsion refers to the use of the technique of successive bursts of thrust at perigee as the vehicle circles the Earth in ever increasing elliptical orbits. The final burst of power sends the vehicle onto its interplanetary trajectory.

There are other reasons for considering low-power nuclear rockets. First of all, development of small nuclear rocket powerplants should be simpler and less costly and therefore shorten an engine development program. The development of the low-power nuclear rocket would be the quickest way of achieving a practical nuclear powerplant. The early availability of the nuclear powerplant would then permit early use of nuclear rockets. Earlier operational and flight experience will be gained to help in the development of larger nuclear rockets. Inasmuch as the development of nuclear rockets would be simpler and less costly, there would be less risk involved in developing such a system.

The purpose of this lecture will be to point out the velocity requirements of low-power missions and to discuss the performance that would be attainable with the use of these low-power missions on booster systems that are currently developed or being developed.

POWERPLANTS FOR LOW-POWER MISSIONS

The previous lecture on low-power nuclear rockets (ref. 8) indicated that there were two reactor powerplants that showed promise of being developed into low-power nuclear rockets. The first was the fast nuclear rocket reactor based on the use of tungsten - uranium dioxide as the fuel-element material, and the second was the water moderated nuclear rocket also using tungsten - uranium dioxide as the fuel-element material. The powerplant weights for fast tungsten - uranium dioxide nuclear rockets that were predicted in reference 8 are shown in figure 1 as functions of reactor power and reactor exit dynamic pressure. This powerplant operates with a reactor exit pressure of 600 pounds per square inch and has a fuel loading of 50-volume-percent uranium dioxide in the tungsten plus uranium dioxide fuel matrix. The minimum powerplant weight occurs for 0 megawatts and is about 1600 pounds. At power levels higher than this the weight increases so that at 1000 megawatts, for example, the powerplant would weigh from 3200 to 3900 pounds, depending on the achievable reactor dynamic pressure. At 500 megawatts the powerplant would weigh about 2300 to 2700 pounds, depending on the achievable reactor dynamic pressure.

The specific impulse that would be attained is a direct function of the reactor exit temperature. The magnitude of the temperature that could be attained from the nuclear rocket powerplant could only be determined after an extensive fuel-element and reactor development program. Figure 2 shows

that for pressure levels greater than 500 pounds per square inch, if a reactor exit temperature of 4500° F could be attained, the specific impulse would be about 890 pounds per pound per second. At 4000° F the specific impulse would be about 820 seconds, at 3500° F about 770 seconds, and at 3000° F about 720 seconds.

Another reactor powerplant weight that showed promise in reference 8 for low-power application was the water-moderated tungsten - uranium-dioxide reactor. The powerplant weight as a function of reactor power and reactor dynamic head is shown in figure 3. Again the reactor exit pressure is assumed to be 600 pounds per square inch. In this case the volume ratio of uranium dioxide and tungsten plus uranium dioxide is assumed to be 15 percent. Inasmuch as the required void area is greater than the flow area because of the necessary provision for voids for insulation and core structure, the void area is assumed to be $1\frac{1}{2}$ times the flow area. The weight of tungsten per unit void area is assumed to be 800 pounds per square foot. For power levels less than 200 or 300 megawatts, the powerplant weight indicated is about 1000 pounds. At 1000 megawatts the powerplant weight varies from 2400 to 3300 pounds, depending on the reactor exit dynamic pressure that may be achievable after the development of the fuel elements. At a power level of 5000 megawatts, the powerplant weight varies from 1500 to 1900 pounds, depending on the achievable reactor dynamic pressure. The specific impulse attainable for this propulsion system is again a function of the outlet reactor temperature that can be achieved. The outlet gas temperature that can be achieved will only be determined after an extensive fuel-element, reactor, and powerplant development program. The specific impulse as a function of reactor outlet temperature is shown in figure 2. The specific impulses assumed for the purpose of performance calculations are purely arbitrary and happen to be the specific impulses that were assumed for each of the studies from which the data were obtained.

MANNED INTERPLANETARY TRIPS

Manned interplanetary trips require that the vehicle leaving the Earth's orbit weigh approximately 2,000,000 pounds. The thrust requirement for this mission on leaving the Earth is approximately 500,000 pounds of thrust, which corresponds to about 10,000 megawatts. Capture at Mars (going into orbit) will require 3000 to 5000 megawatts. At this point a large fraction of the vehicle mass would have been expended. In addition, mass is left on the Mars surface and in the Mars orbit; propellant and supplies are consumed; etc. On leaving Mars the reactor power required would therefore be reduced to the order of 1000 megawatts. This would be classed as a low-power application. Capture at the Earth would require less than 1000 megawatts (again a use for low-power nuclear rockets).

Low-power nuclear rockets could be applied to the Earth escape portion of a manned interplanetary mission if the technique of perigee propulsion is used. This technique (ref. 9) is illustrated in figure 4. The vehicle starting from the low lying Earth orbit is given successive bursts of thrust at perigee. The thrust period occurs over angle $\Delta\theta_1 + \Delta\theta_2$. After each burst of thrust the vehicle will lie in an elliptical orbit with a greater apogee than previously. An arbitrary number of perigee bursts are given prior to a final continuous thrust phase. Figure 4 from this reference illustrates the trajectories for a case where the thrust to weight ratio is 0.03 and the hyperbolic velocity is 3 miles per second. The mass ratio (residual load to initial weight ratio) attained for different numbers of perigee propulsion thrust bursts prior to the final continuous thrust period is shown in figure 5. The case for one propulsion segment corresponds to continuous thrust all the way at a thrust to weight ratio of 0.03. Two propulsion segments correspond to one burst followed by one continuous thrust period. Likewise 3, 4, and 5 correspond to 2, 3, and 4 bursts, respectively, prior to a final continuous thrust period. The case of nine propulsion segments corresponds to eight bursts followed by a ninth continuous thrust period. In this case the mass ratio that is achieved corresponds almost to that achieved with an impulsive type thrust. Inasmuch as the vehicle circles the Earth several times prior to departure, the total time elapsed over the entire perigee propulsion period is greater than that required for continuous thrust. For example, in the case of nine propulsion segments approximately 5 to 6 days are required to accomplish the final velocity increment instead of a fraction of a day required for continuous thrust. The operating time of the powerplant is also increased by use of perigee propulsion. This increase is roughly inversely proportional to the thrust to weight ratio. In other words, if the thrust to weight ratio is 0.03 in place of 0.3, then approximately 10 times the operating time will be required for the engine. In addition, the engine must be started and stopped several times; in this particular case, nine times. For manned interplanetary missions the increased total elapsed time for Earth escape is not significant when compared to the 400 to 500 days required to complete the mission. The only factors that are significant is that the engine operating time must be increased and that the engine must be capable of being recycled.

The performance potential available by the use of perigee propulsion is compared with continuous thrust in figure 6. The residual load ratio, that is, the final residual load over the initial gross weight, is plotted as a function of the initial value of the thrust to weight ratio. Also shown is a scale that indicates approximate reactor power for a vehicle that weighs 500,000 pounds. The mission considered is a 209-day trip to Mars. In this case, the hydrogen outlet temperature was assumed to be 4500°F . In the case of perigee propulsion, the specific impulse associated with 4500°F would be higher than that for the continuous thrust case because the reactor could be operated at a lower pressure, which yields a higher specific impulse as indicated in a previous figure. The continuous thrust case requires

a thrust to weight ratio of about 0.3 before the residual load ratio levels off and no further gain is obtained by further increases in thrust to weight ratio. For a thrust to weight ratio of 0.03, the mass ratio of perigee propulsion is slightly greater than the continuous thrust case because of the higher specific impulses attainable when operating the reactor at a lower pressure. It is possible therefore to use the technique of perigee propulsion and to obtain reactor power or thrust levels that are approximately 1/10 that required for continuous thrust. The powerplant must be capable, however, of being operated for periods of 10 times longer than continuous thrust powerplants and must also be capable of being recycled of the order 5 to 10 times. The implication here is that a single 1000-megawatt reactor could be used to power an entire manned Mars mission of 2,000,000 pounds initial starting weight in Earth's orbit without any sacrifice in performance over the system where a 10,000-megawatt reactor is to be used.

Unmanned Pioneering Probes

Unmanned pioneering probes are those probes that are the forerunner of manned vehicles. Their function is to determine the environment that man will be exposed to on interplanetary voyages and to gather necessary data to determine the protection that will be required by man for these trips. Typical missions include (1) solar probes, (2) planetary atmosphere and field probes, (3) planetary surface probes, and (4) interplanetary space environment probes. The purpose of the solar probe is to gather data that might be useful in predicting solar flares. These probes would be used to measure and observe conditions on the surface of the sun in order to make interplanetary space weather predictions.

One-way planetary atmosphere and field probes would be used to determine the atmospheric conditions existing on the target planets and in addition to determine the extent of any radiation belts or other fields that may exist around the planets. In other words, we would want instrumented probes to follow the trajectories that men would follow at a future date so that unknowns will be discovered by probes rather than man. Planetary surface probes will explore the surface of the planet to determine the conditions that exist on the surface and find conditions that may be hostile to man so that adequate provisions can be made to protect man when he does land on these planets. Interplanetary space environment probes will be used to find the conditions that exist in the space between Earth and the planets to be visited. For example, the extent of meteorite activity will be determined along the trajectories that might be used for manned voyages.

UNMANNED SCIENTIFIC PROBES

Unmanned scientific probes are those missions for the purpose of scientific exploration of space in which man cannot be included. Such missions

include (1) solar probes, (2) one-way planetary probes, (3) solar system exploration, (4) exploration beyond solar system, and (5) out-of-ecliptic space probes.

The solar probes would be used for scientific measurements of the sun, which are required to obtain a more thorough understanding of the processes that occur in solar flare activity, to measure magnetic and gravitational fields about the sun, and to obtain any other information that is deemed necessary in understanding the center of our solar system. One-way planetary probes offer the possibility of sending much larger payloads to the vicinity of any planet that is of interest, inasmuch as the payload would not have to be returned as would be the case when men are included. These probes permit the use of smaller boosters for a given payload, or for a given booster they would permit a much larger payload to be sent to the planet.

In the case of planets that are at the extremities of the solar system, large velocity increments are required to have the payloads achieve their objective in a reasonable amount of time. One-way probes would permit such missions to be accomplished in the shortest possible time when compared to the two-way probes.

Similar advantages occur for one-way probes for solar system exploration. High-velocity increments are required in order to reach the extremities of the solar system in reasonable times. Solar system exploration would be useful for gathering data on meteorite density and the extent of magnetic, electrical, and gravitational fields that exist in the solar system, all of which would be important in helping determine the origin of our solar system.

As in the case of exploration at the extremities of our solar system, exploration beyond our solar system is beyond the capabilities of any systems if man were included. Probes beyond the solar system say at approximately 70 astronomical units would be useful in determining meteorite densities, the source of meteorites, and the study of comets and intergalactic fields at this large solar system radius. The fields due to the sun would be a minimum, and it would be possible perhaps to measure fields that are due to other galaxies or fields that exist in the space between galaxies.

Another interesting scientific mission is one of studying the solar system space outside of the ecliptic plane. There are indications that meteorite density is a maximum in the plane of the ecliptic. Should it be true that the meteorite density is low outside the ecliptic plane, it may be worthwhile to use out-of-ecliptic trajectories for manned interplanetary missions to minimize contact with meteorites.

The missions discussed serve to illustrate the large variety of missions that are of interest in the scientific exploration and manned exploration.

tion of space. It would be highly desirable to have vehicles available that could accomplish these missions. In fact, if vehicles that could accomplish these missions were available, the scientific community would surely find many other uses for these systems. They would broaden the scope of what the scientist could explore.

VELOCITY INCREMENT REQUIREMENTS

The velocity increment requirements for the various missions discussed are shown in tables I and II. Table I lists the velocity requirements for solar probe missions. For the purpose of the data on this table it was assumed that all the probes were launched from a 300-mile orbit about the Earth, and the vehicles had an initial thrust to weight ratio of 0.3. The perihelion radius, the aphelion radius, and the inclination to the ecliptic plane are used to describe the various solar missions. The ΔV corresponding to each combination of these three parameters is shown in the last column in miles per second. For example, the solar flyby probe with a perihelion of 0.3 astronomical unit requires a 4.2 mile per second velocity increment. Decreasing the perihelion radius to 0.025 astronomical unit requires a ΔV of 11.2 miles per second. For some observations of the sun it may be necessary to provide a probe that orbits the sun at various perihelion and aphelion radii. In order to orbit the sun in an orbit that has a perihelion of 0.3 astronomical unit and an aphelion of 0.7 astronomical unit would require a velocity increment of 6.2 miles per second. For a circular orbit at 0.3 astronomical unit a ΔV of 12.4 miles per second is required. A stationary orbit about the sun occurs at a radius of 0.167 astronomical unit. This particular probe would require a ΔV of 18.1 miles per second. Solar orbiters that are inclined to the ecliptic plane are shown as the last group of missions on this table. Each of these orbiters have a circular orbit radius of one astronomical unit. For an inclination of 20° to the ecliptic plane a velocity increment of 4.6 miles per second is required. The velocity increment increases progressively from 7 to 10.9, to 14.9, to 22.3 miles per second as the inclination is increased from 30° , to 45° , to 60° , to 90° .

Velocity requirements for planetary and deep space probes are shown in table II. The thrust to weight ratio of the vehicles that are launched from a 300-mile orbit is 0.3. The trip time and ΔV for the various missions are shown. For solar system escape a ΔV of 5.5 miles per second beyond the 300-mile orbit is required. The Venus flyby in 75 days requires 2.6 miles per second. The Mars flyby requires 3.6 miles per second for a 100-day trip. The Jupiter flyby requires 5.3 miles per second. A Mars orbiter requires 7.4. A Mars round trip, which was computed for atmospheric breaking at Earth in 1980 and with a 40-day wait time, requires a velocity increment of 8.9 miles per second for a 420-day trip. A 460-day Venus round trip, again with atmospheric breaking at Earth and with a 40-day wait time at the planet, requires a ΔV of 7.3 miles per second.

The ΔV requirement for a one-way probe to a distance of approximately twice the radius of the solar system, which corresponds to about 70 astronomical units, is shown in the next three missions. To accomplish this mission in 30 years requires a ΔV of 5.8 miles per second. Reducing the time to 20 years and to 10.2 years increases the velocity requirement from 5.8 to 6.5, to 10 miles per second, respectively.

The final mission shown on this table is a 70-astronomical-unit orbiter. In order to accomplish this mission in 30 years requires a ΔV of 10.1 miles per second, in 20 years it requires 13.5 miles per second.

It will be noticed that many of the missions that have been discussed in tables I and II required ΔV 's beyond 10 miles per second, some in fact go as high as 22 miles per second. These are truly high-energy missions when compared, for example, to the solar system escape ΔV of only 5.5 miles per second.

PERFORMANCE ESTIMATES FOR UNMANNED MISSIONS

In making performance estimates for the low-power nuclear rocket, missions using the boosters currently developed or being developed were considered to place the nuclear stages into 300-mile orbits (see ref. 4). The boosters considered are the Atlas-Centaur, the Saturn-IB, and the Saturn-V. The Atlas-Centaur is assumed to be capable of placing a nuclear stage weighing 9000 pounds into a 300-mile orbit. The Saturn-IB and Saturn-V are assumed to be capable of placing nuclear stages weighing 28,500 and 220,000 pounds, respectively, into a 300-mile orbit. The thrust level required for nuclear stages atop these boosters is shown in table III based on the assumption that a thrust to weight ratio of 0.3 is optimum for orbital launch vehicles. The thrust level is merely the stage weight multiplied by the thrust to weight ratio, which is 0.3. The Atlas-Centaur has a thrust level of 2700 pounds, which corresponds to a power level of 60 megawatts. The Saturn-IB nuclear stage would require a thrust level of 8600 pounds, which corresponds to a power of 180 megawatts. The Saturn-V nuclear stage would require a thrust of 66,000 pounds, which corresponds to a power of 1300 megawatts.

If perigee propulsion is used to power these orbital launch nuclear stages, the thrust and power levels can be reduced by a factor of 10 without affecting the performance. The highest reactor power required for nuclear stages for the boosters that are currently being developed in the United States space program require reactor powers that are no higher than 1300 megawatts. If perigee propulsion is used, they require powers no higher than 130 megawatts.

Atlas-Centaur Nuclear Stage

The performance capability of a nuclear stage placed into orbit by the Atlas-Centaur is shown in figure 7. The nuclear stage in this case weighs 9000 pounds and is launched from a 300-nautical-mile Earth orbit. The figure shows the velocity increment beyond this 300-nautical-mile orbit as a function of payload. The performance of a single nuclear stage powered with a 60-megawatt water moderated tungsten-184 reactor is shown by the upper curve. For comparison, the performance of a one-stage chemical hydrogen-oxygen system is also shown. For payloads of 1000 pounds or less, the Atlas-Centaur boosted nuclear stage should be able to achieve velocity increments from 6 to 8 miles a second, which is about $2\frac{1}{2}$ miles per second more than could be accomplished with a chemical hydrogen-oxygen stage.

Saturn-IB Nuclear Stage

The performance of one- and two-stage nuclear vehicles boosted into a 300-nautical-mile orbit by Saturn-IB is shown in figure 8. The one- and two-stage vehicles have an initial weight of 28,500 pounds, which is the capability of the Saturn-IB placing a payload into a 300-nautical-mile orbit. The velocity increment beyond the 300-nautical-mile orbit in miles per second is plotted as a function of payload in pounds. Curves for chemical hydrogen-oxygen systems are also shown in this figure for comparison purposes. The solid lines in both the chemical and nuclear vehicles correspond to a single-stage operation. The dashed lines correspond to two-stage operations.

Considering first single-stage operations, the nuclear stage requires a power level of 180 megawatts and will produce velocity increments that are on the order of 3 or 4 miles per second greater than the chemical hydrogen-oxygen stage. For a payload of 4000 pounds or less, it is possible to achieve velocity increments from 6 to 9 miles per second in the case of the nuclear stage. For a payload of 1000 pounds, the velocity increment of almost 8 miles per second could be achieved. A two-stage nuclear vehicle, which had 180-megawatt power for the first stage and 60-megawatt power for the second stage, would increase the velocity increment attainable for a 1000-pound payload to about $9\frac{1}{2}$ miles per second.

Saturn-V Nuclear Stage

The performance of nuclear stages boosted into a 300-nautical-mile orbit by the Saturn-V is shown in figure 9. The initial weight of both the one-stage and two-stage vehicles is 220,000 pounds. Chemical hydrogen-oxygen one- and two-stage vehicles are shown for comparison with the nuclear one-

and two-stage vehicles. The nuclear one-stage vehicle has a power of 1300 megawatts. The two-stage nuclear vehicle has a power of 1300 megawatts in the first stage and 180 megawatts in the second stage. The nuclear vehicles produce about 4 miles per second greater velocity increment than the hydrogen-oxygen stages. For a payload of 4000 pounds, the single-stage nuclear vehicle can achieve a velocity increment of about 9 miles per second. Two stages increase the velocity increment to about 13 miles per second.

Maximum Potential

In order to indicate the maximum ΔV potential of one- and two-stage nuclear rocket vehicles, starting from a 300-nautical-mile Earth orbit, the velocity increment was plotted as a function of initial weight in orbit and extended beyond the capability of the Saturn-V. In figure 10 it can be seen that the one-stage nuclear vehicle approaches asymptotically a velocity increment of about 10 miles per second. The two-stage nuclear vehicle approaches a velocity increment of about 16 miles per second. In addition, it can be noted that not much gain is to be had by increasing the initial weight in orbit beyond the capability of the Saturn-V if we are limited to one- and two-stage nuclear velocities.

Although all the performance estimates were made using the tungsten water-moderated reactor, similar performance would be shown for the fast reactor, except perhaps for the case of the Atlas-Centaur boosted nuclear stage. At the power levels required for the Atlas-Centaur nuclear stage, the fast reactor was calculated to have a weight of approximately 1000 pounds more than that for the water-moderated system. By referring to figure 7, it is seen that the zero-payload point for the fast reactor nuclear stage boosted by the Atlas-Centaur would correspond to a 1000-pound payload case for the water-moderated system since the difference in their weight is 1000 pounds. The fast reactor would still exhibit a 1/2 mile per second greater performance capability than the chemical hydrogen-oxygen system.

EFFECT OF POWERPLANT OPERATING CONDITIONS ON PERFORMANCE

For all the calculations that have been made on the thrust and weight of powerplants, it was assumed that the reactor operating pressure was 600 pounds per square inch. A study has been made in reference 10 to determine what the best reactor operating pressure should be. Increasing the pressure has the effect of increasing the pressure shell weight at the same time as the core weight is reduced. Eventually this leads to an increase in overall powerplant weight as pressure is increased. On the other extreme, as the reactor pressure is decreased the reactor core becomes heavier because it is larger in size, but the specific impulse increases because of the lower operating pres-

sure. The balance between these two when an entire mission is considered produces a best operating pressure. Data taken from reference 10 are plotted in figure 11. The payload for a Mars mission is plotted as a function of reactor exit pressure for a nuclear rocket stage weighing 30,000 pounds. A range of reactor operating temperatures are shown from 3500° to 4500° F. Three trip times are considered: 120 days, 170 days, and 259 days. Also two systems are considered: one is a pump system shown by the solid curve, and the second is a pressurized system indicated by the dashed curve. The optimum in reactor exit pressure occurs for the following reason. As the pressure is increased the reactor core is reduced in weight; however, the pressure shell weight, the turbopump weight, and the nozzle weight increase due to the higher pressure. The net effect is that if pressure is increased too high the powerplant weight will increase. On the other hand, reducing pressure level increases the reactor weight, but at the same time the specific impulse increases at the lower operating pressures as shown by the curve in figure 2. The net effect is that there is a best operating pressure somewhere in between the two extremes. In the case of the pumped system, the optimum pressure occurs somewhere in the range of 100 to 200 pounds per square inch. It should be noted, however, that the curves are quite flat and there is a large leeway in the pressure that could be chosen. In the case of the pressurized system, the optimum pressure occurs at a much lower value being on the order of 20 to 30 pounds per square inch. This, of course, is due to the fact that if we tried to operate the pressurized system at higher pressures the hydrogen tank weight becomes prohibitive, thus shifting the optimum to the left as it has done in the figure.

It should be noted that if shielding is necessary it is included in the payload of figure 14. If the shield weight is significant when compared with the powerplant, its variation with reactor size should be taken into account in the optimization. Because the low operating pressures go along with larger reactors, the shield weight would tend to be proportionately greater for low-pressure powerplants than for high-pressure powerplants. The optimums would therefore tend to shift toward the higher pressure somewhat, depending on how much shielding is required.

It is interesting to note that the use of the pressurized system has reduced the payload but by only a small amount. For example, at 4500° F in the 170-day case, the payload has been reduced by approximately 10 percent in going from the pump to the pressurized system. It may be desirable to eliminate complexity in favor of a slight penalty in payload. In addition, the probability of success of a reactor development program would probably be greater for the pressurized system, inasmuch as the power densities would be greatly reduced. In fact, another possibility is that because of the lower power density and the lower dynamic heads required in pressurized systems, it may be possible to operate materials at a higher temperature level than in the case of pump-fed high-density systems with resultant increase in specific impulse. This is not reflected in the curves shown in this figure; consequently, the penalty is, at most, the penalty

shown here. There appears to be about a 20-percent decrease in payload for a 1000° F reduction in outlet temperature. If the pressurized system could be made to operate at a temperature level 500° F higher than the pump system, the penalty in payload in going to a pressurized system would be eliminated. Based on the performance shown in this curve, it would be well to investigate pressurized versus pump systems further.

SUMMARY

The large velocity increments necessary to achieve scientific missions of interest require the use of lightweight nuclear powerplants since all chemical systems cannot accomplish many of the missions which the nuclear systems can. Both one- and two-stage nuclear rockets were shown to have significantly superior performance over the best chemical systems. It will always pay to add a final nuclear stage to any system regardless of whether the booster is as small as the Atlas-Centaur or as large as the Nova, providing that the nuclear powerplant for these stages can be made as light in weight as the calculations in references 4, 5, and 8 have indicated.

Inasmuch as the current United States space program is concerned chiefly with achieving the Apollo moon mission, much less emphasis is placed on the future manned interplanetary missions. In addition, since the Apollo mission and the probe missions that are currently contemplated in the space program can be accomplished with chemical vehicles, the nuclear rocket program is of secondary importance. Since experience has shown that any new engine development is very costly, great caution is exercised in starting new development programs. If developed nuclear rocket engines were on hand they would surely be used in the space program. The fear of the great cost involved in developing these powerplants has caused the nuclear rocket program to be directed toward the development of only one powerplant, that is, a large power nuclear rocket that will be required for future manned interplanetary missions. There is no plan to develop small lightweight nuclear rockets in the current United States program. Perhaps this is an area where other NATO nations may be able to make a contribution to the space program.

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TABLE I. - VELOCITY REQUIREMENTS FOR SOLAR PROBES

[Thrust to gross weight ratio, 0.3; launch from
300-mile orbit.]

| Mission | r_p , AU | r_a , AU | Inclination to ecliptic plane, deg | ΔV , mps |
|---------------|---------------|---------------|--|---------------------|
| Solar flyby | 0.3 | ----- | 0 | 4.2 |
| | .2 | ----- | 0 | 5.5 |
| | .1 | ----- | 0 | 7.8 |
| | .025 | ----- | 0 | 11.2 |
| Solar orbiter | 0.3 | 0.7 | 0 | 6.2 |
| | .3 | .5 | 0 | 8.4 |
| | .3 | .3 | 0 | 12.4 |
| | .2 | .7 | 0 | 7.4 |
| | .1 | .7 | 0 | 9.5 |
| | .167 | .167 | 0 | 18.1 |
| | | | (Stationary) | |
| | 1.0 | 1.0 | 20 | 4.6 |
| | 1.0 | 1.0 | 30 | 7.0 |
| | 1.0 | 1.0 | 45 | 10.9 |
| | 1.0 | 1.0 | 60 | 14.9 |
| | 1.0 | 1.0 | 90 | 22.3 |

TABLE II. - VELOCITY REQUIREMENTS FOR PLANETARY AND DEEP SPACE PROBES

[Thrust to gross weight ratio, 0.3; launch from 300-mile orbit.]

| Mission | Time, days | ΔV |
|---|---------------|------------|
| Solar system escape | --- | 5.5 |
| Venus flyby | 75 | 2.6 |
| Mars flyby | 100 | 3.6 |
| Jupiter flyby | 413 | 5.3 |
| Mars orbiter | 125 | 7.4 |
| Mars round trip (atmospheric braking at Earth; 1980; 40-day wait time) | 420 | 8.9 |
| Venus round trip (atmospheric braking at Earth; 40-day wait time) | 460 | 7.3 |
| Flyby probe beyond Pluto (70 AU) | 30 (yr) | 5.8 |
| | 20 (yr) | 6.5 |
| | 10.2 (yr) | 10.0 |
| 70 AU orbiter | 30 (yr) | 10.1 |
| | 20 (yr) | 13.5 |

TABLE III. - NUCLEAR STAGE REQUIREMENTS

| Booster | Mission starting point | Nuclear stage | | |
|---------------|--|---------------|---------------|--------------|
| | | Weight, lb | Thrust, lb | Power, Mw |
| Atlas-Centaur | Low-altitude orbit (continuous thrust) | 9,000 | 2,700 | 60 |
| Saturn-IB | | 28,500 | 8,600 | 180 |
| Saturn-V | | 220,000 | 66,000 | 1300 |
| Atlas-Centaur | Low-altitude orbit (perigee propulsion) | 9,000 | 270 | 6 |
| Saturn-IB | | 28,500 | 860 | 18 |
| Saturn-V | | 220,000 | 6,600 | 130 |